NATIONAL ADVISORY COMMITTEE

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 3258

INVESTIGATION OF MACH NUMBER CHANGES OBTAINED BY DISCHARGING HIGH-PRESSURE PULSE THROUGH WIND TUNNEL OPERATING SUPERSONICALLY

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SUMMARY

A series of tests was performed to obtain an indication of the transient-flow phenomena caused by discharging a chamber of high-pressure gas into a wind tunnel operating supersonically. For the configurations tested, two types of gust were obtained. One had a maximum Mach number with a practically zero time duration. The other had a maximum Mach number with a finite time duration depending on the specific geometry. Such configurations are applicable as supersonic longitudinal-gust tunnels.

INTRODUCTION

When the diaphragm of a shock tube is ruptured, a plane-shock wave and a temperature-discontinuity (contact) surface are propagated into the low-pressure chamber. Behind the shock the magnitude of the flow velocity depends upon the kinds of gas on each side of the diaphragm, their pressures, and their temperatures (ref. 1, e.g.). If strong shocks propagate into moving air, even though initiated in air locally at rest, substantial increases in the Mach number of the moving air can theoretically be realized. Hence, by suitably passing shock waves into a supersonic-tunnel stream, instantaneous changes in the flow Mach number might be obtained, thus providing a laboratory facility for generating longitudinal gusts in supersonic flow. These gusts would introduce discontinuities in both pressure and velocity, and therefore simulate flight through a blast or shock wave. Such a facility would be of value for experimental investigations of transient phenomena, particularly in the field of aircraft dynamics.

The present experimental investigation was undertaken to obtain an indication of the Mach number changes that could be realized by initiating a high-pressure pulse near the upstream end of a supersonic-wind-tunnel nozzle and permitting the pulse to propagate downstream into the

supersonic flow. Because the resulting flow is influenced by factors such as complex shock reflections and wakes, it is not readily amenable to mathematical analysis. These tests were performed at the NACA Lewis laboratory.

APPARATUS

The tests were performed in a 3.4- by 3.4-inch duct. During the course of the investigation, three different nozzles were used. Two were two-dimensional conventional supersonic nozzles designed for Mach numbers 1.6 and 1.9 and the third was an axially symmetric multinozzle. The arrangements of the apparatus are shown in figure 1. The geometry and some characteristics of the steady flow obtained with the multinozzle are reported in reference 2. For the present tests a $1\frac{1}{2}$-inch-diameter hole was cut through its center so that gas from the high-pressure cylinder could enter the tunnel (fig. 2). The steady-flow Mach number with the multinozzle as used in the present tests was 1.6.

Two high-pressure cylinders were fabricated from $1\frac{1}{2}$- and 3-inch-diameter steel pipe and were each about 4 feet long. One end of these cylinders was capped, whereas the other was fitted with a flange clamp to hold a metal (brass shim stock) diaphragm. The pipe was strut supported within the 10-inch-diameter inlet pipe of the duct. Because of space limitations, the high-pressure cylinder was curved approximately 2 feet from its downstream end, as indicated in figure 1. The diaphragm was positioned either within the bellmouth of the conventional nozzles or at the upstream face of the multinozzle centered on the $1\frac{1}{2}$-inch-diameter hole. It was ruptured by a manually operated sharp-pointed plunger. Dry commercial nitrogen was used as the high-pressure gas.

For several tests with the Mach number 1.9 conventional nozzle, a shroud was installed between the diaphragm and the bellmouth (figs. 1 and 3). This shroud was perforated with 3/8-inch-diameter holes to permit passage of the air for steady operation of the wind tunnel. The ratio of open area to total surface area of the shroud was about 0.55.

A 5° half-angle wedge was mounted in the test section so that its shock pattern could serve as an indication of Mach number. Schlieren photographs of the flow in the neighborhood of this wedge were recorded with a high-speed motion-picture camera. The wind tunnel was operated with atmospheric inlet pressure and a stagnation temperature of about 120° F.

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RESULTS AND DISCUSSION

Two-Dimensional Conventional Nozzles

Schlieren photographs of the 5° wedge in the test section are presented in figure 4 for the configuration of figure 1(a) with the diaphragm $13\frac{1}{4}$ inches upstream of the nozzle throat and with the $1\frac{1}{2}$-inch-diameter high-pressure cylinder. The pressure ratio across the diaphragm was 0.072. The time interval between schlierens in figure 4 was about 1/4 millisecond. The photographs show the decrease in shock angle after the diaphragm was ruptured and the shock wave (not visible) passed downstream (cf. frames 4 and 8, e.g.). The stream Mach number (estimated to tenths of Mach numbers by measuring shock angles) increased from 1.9 to 2.4 instantaneously, then decreased to its original value within less than 3 milliseconds. About 25 milliseconds later (not shown in fig. 4), pieces of the ruptured diaphragm appeared, following which steady tunnel flow reoccurred.

Similar photographs were obtained at other pressure ratios, with the diaphragm initially closer to the throat, and with the Mach number 1.6 nozzle. All the results obtained with the configuration of figure 1(a) are summarized in table I(a).

There was the possibility that the rarefaction wave, which reflected from the closed end of the high-pressure chamber, overtook and weakened the shock. In addition, it was believed that the strength of the initial shock was attenuated greatly by spherical spreading at the end of the high-pressure cylinder, partial propagation upstream into the inlet pipe, and by shock reflections in the wind-tunnel nozzle.

A shroud (fig. 3) was therefore constructed to eliminate the area discontinuity between the diaphragm and the bellmouth and thus, presumably, to reduce shock attenuation. Also, the high-pressure-pipe diameter was increased to 3 inches so that a larger mass of nitrogen would be expelled. The results, for the shroud with the diaphragm 11 inches upstream of the tunnel throat, are summarized in table I(b). There was no increase in the Mach number behind the shock as compared with the results obtained without the shroud.

For these tests, both with and without the shroud, the results are interpreted as follows: The air just downstream of the diaphragm constitutes the wake of a bluntly terminated body (the high-pressure cylinder) and is essentially at rest relative to the body. When the diaphragm is ruptured, a shock wave is initiated in this wake. The initial pressure ratio across the shock is precisely that calculated from shock-tube theory in terms of initial pressure ratio across the diaphragm. However, as the shock propagates out of the still-air region through the

mixing zone and into the main stream, its strength and contour must change. The flow acceleration and streamline curvature in the nozzle, together with shock reflections from the walls, further change the strength of the main shock. The final strength of the shock as it passes through the test section is not related in any simply calculable way to the initial pressure ratio across the diaphragm.

If now the (unknown) pressure ratio of this final shock is taken as an independent variable, it is possible to calculate the Mach number behind the shock as a function of pressure ratio for fixed values of the Mach number ahead of the shock. Some results of such calculations are shown in figure 5. The curves show that the increase in the Mach number behind the shock, at a given shock pressure ratio, becomes less as the initial Mach number becomes larger (at Mach number 5 and above, the Mach number behind the shock is always less than its initial value). Furthermore, the shock strength for the maximum Mach number M_b is finite rather than infinite $(P_a/P_b = 0)$ as for the ordinary shock tube $(M_a = 0)$.

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The maximizing depends upon the rise in the speed of sound (temperature) across a shock wave; above a certain shock strength this rise more than offsets the speed increment added by the shock so that the Mach number behind the shock decreases from its maximum value. For an initial Mach number M_a of 1.6, the maximum Mach number behind the shock M_b is 2.35; whereas for an initial Mach number of 1.9, the maximum attainable Mach number is 2.53. Within the limits of experimental accuracy, these maximums were never exceeded (cf. table I).

In an ordinary shock tube a contact surface follows the shock into the low-pressure chamber after the diaphragm is ruptured. Although the velocity and pressure are the same on either side, the temperature is lower behind the contact surface. The Mach number behind this temperature discontinuity is therefore greater than that ahead of it. In the present tests changes in the Mach number corresponding to the passage of the contact surface were not observed. This is attributed to rapid mixing of the flow in the bellmouth of the tunnel and to reflected shocks in the nozzle, which obliterated the contact surface.

As the shock passes through the nozzle and test section, a phenomenon occurs similar to the nonstationary starting phase of a wind tunnel (ref. 3). During this interval, reflected oblique shocks caused by the passage of the main shock occur in the nozzle. In the test section the Mach number first increases abruptly as the shock passes, as

indicated qualitatively in sketch 1, and then attenuates to the steady-state value. Thereafter the tunnel operates at the steady-state Mach

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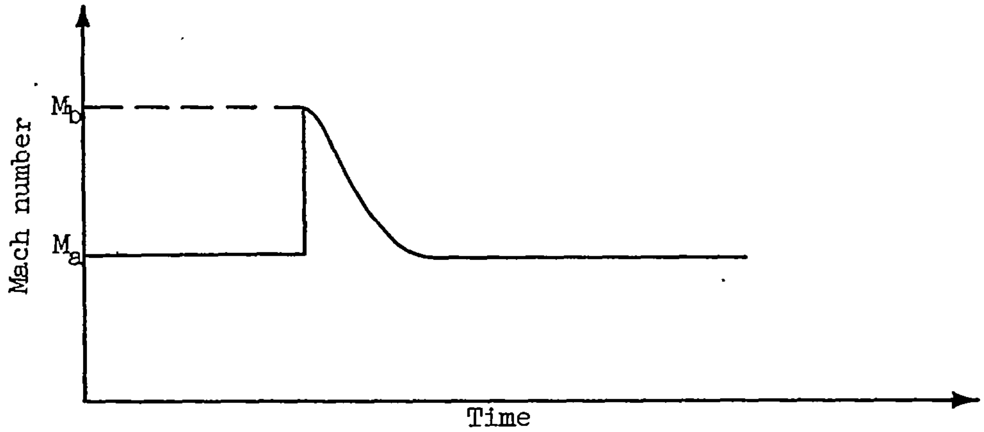


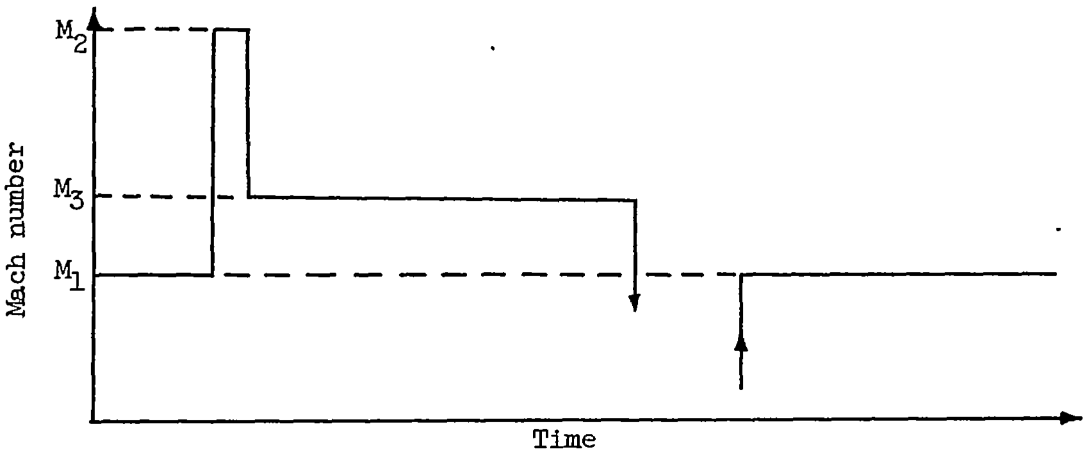
Sketch 1

number, although temporarily, perhaps, at a larger stagnation pressure than the initial pressure. As the reservoir gas is depleted, the stagnation pressure decreases until normal tunnel inlet pressure is reached.

Axially Symmetric Multinozzle

A final series of tests was made using the 3/4-inch-thick multinozzle of reference 2 to establish the supersonic flow prior to the diaphragm rupture. The steady flow at Mach number 1.6 contained many oblique shocks originating at all the edges of the multinozzle.

With a pressure ratio of 0.048 across the diaphragm, a Mach number of 4.0 with a duration of 1/2 millisecond was obtained after rupture. This is shown by the schlieren photographs of figure 6. Then the flow remained steady at Mach number 2.2 for about 50 milliseconds, except when disturbed by fragments of the diaphragm. This sequence of events is indicated qualitatively in sketch 2.



Sketch 2

Although not apparent in the sketch, the flow through the multinozzle ceased immediately after the diaphragm rupture because of the resultant high pressure on the downstream face of the multinozzle; the Mach number 4.0 pulse was thus entirely a shock-tube flow. The unsteady flow may be directly compared with that discussed in reference 4, for which a shock-tube flow was expanded into a larger-area duct to obtain very large Mach numbers. For the present tests, however, the velocity of the primary (initial) shock, when it reaches the test section, is dependent on both its pressure ratio and the velocity of the gas into which it is passing. 3289

As shown in reference 4, a secondary shock is formed by shock reflections due to the passage of the primary shock through the changing-area section. The Mach number 4.0 flow M_2 occurs between the secondary shock and the contact surface (a temperature discontinuity surface separating the gas originally inside and outside of the high-pressure cylinder). The gas in this region was originally between the diaphragm and the start of the changing-area section (downstream face of the multinozzle). It is inferred from the results of reference 4 that increased time duration of the high Mach number flow should be obtained by increasing the distance between the diaphragm and the downstream multinozzle face. The Mach number of the flow behind the secondary shock is dependent only on the diaphragm pressure ratio and the high-pressure-chamber to test-section area ratio.

The Mach number 2.2 flow M_3 contains the gas originally in the high-pressure cylinder. This flow does not expand isentropically through the diverging-area section, but is affected by wave reflections required to satisfy the static-pressure and the velocity boundary conditions across the contact surface. In addition, this flow is also affected by the oblique shocks created by the impingement of the flow against the tunnel walls (fig. 6). When the pressure of the gas issuing from the high-pressure chamber decreases sufficiently, the supersonic flow over the wedge collapses (indicated by the gap in sketch 2). Supersonic flow M_1 is reestablished when the pressure downstream of the multinozzle becomes low enough to allow restarting of the multinozzle.

CONCLUDING REMARKS

When a shock caused by discharging a chamber of high-pressure gas into the bellmouth of a supersonic tunnel is allowed to pass downstream through the tunnel, an abrupt increase in Mach number is obtained in the test section. This increase is immediately followed by a less abrupt decrease toward the initial value. The maximum Mach number obtainable is limited by the physical properties of the gas.

In a different arrangement, with the pulse entering the tunnel at the downstream face of a multinozzle, a considerable increase in Mach number is obtained. The duration of the larger Mach number flow is finite and can be controlled by adjusting the distance between the diaphragm and the downstream multinozzle face. The Mach number rise is dependent upon the diaphragm pressure ratio and the high-pressure-chamber to test-section area ratio.

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Any of the arrangements tested appear to be applicable as a supersonic longitudinal-gust tunnel for producing transient boosts in Mach number. The choice would be governed largely by the desired shape of the curve of Mach number against time.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, June 17, 1954

REFERENCES

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3. Bull, G. V.: Investigation into the Operating Cycle of a Two-Dimensional Supersonic Wind Tunnel. Jour. Aero. Sci., vol. 19, no. 9, Sept. 1952, pp. 609-614.

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TABLE I. - MACH NUMBERS OF FLOW

(a) Without shroud.

Distance from diaphragm to nozzle throat, in.	Pressure ratio across diaphragm	Mach number at wedge	
		Ahead of shock	Behind shock
$13\frac{1}{4}$	0.072	1.9	2.4
$13\frac{1}{4}$	.048	1.9	2.2
$13\frac{1}{4}$	.048	1.6	2.0
$9\frac{5}{8}$	.072	1.6	2.3
$9\frac{5}{8}$	.053	1.6	2.3
$5\frac{1}{4}$	.072	1.9	2.6
$5\frac{1}{4}$	.072	1.6	2.1
$5\frac{1}{4}$	.048	1.6	2.1

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(b) With shroud.

Distance from diaphragm to nozzle throat, in.	Pressure ratio across diaphragm	Mach number at wedge	
		Ahead of shock	Behind shock
11	0.072	1.9	2.2
11	.053	1.9	2.5

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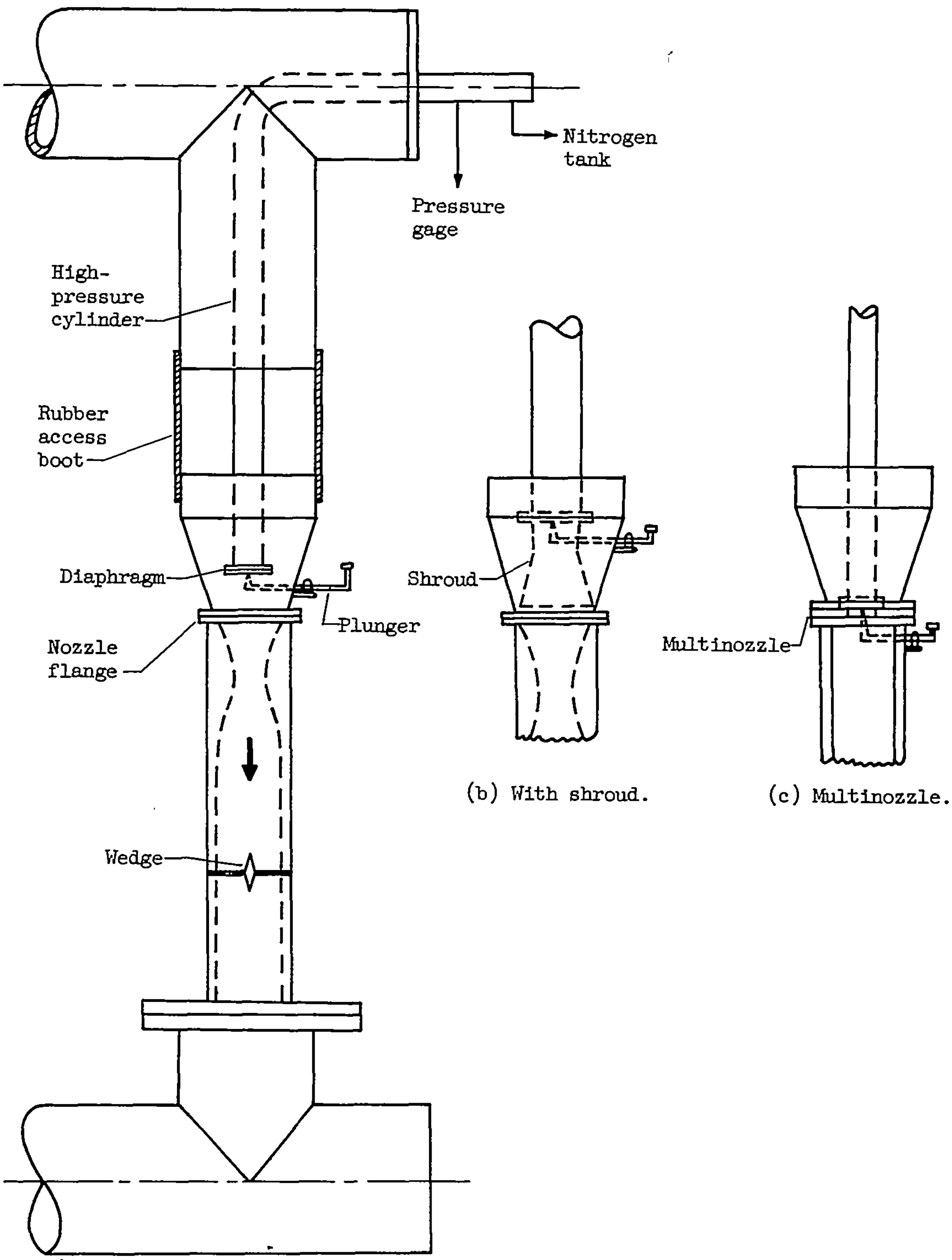

(a) Without shroud.

(b) With shroud.

(c) Multinozzle.

Figure 1. - Test configurations.

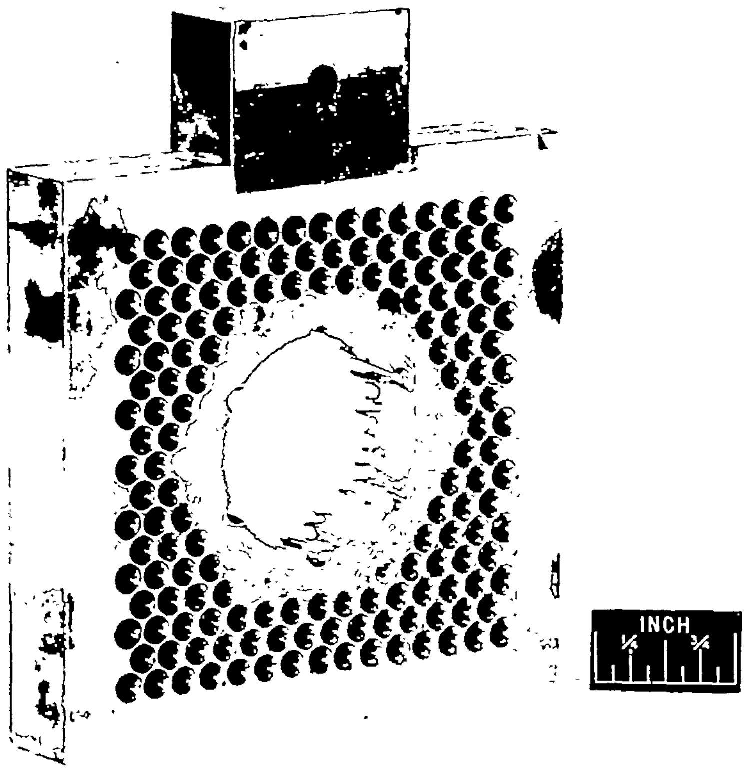

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Figure 2. - Multinozzle.

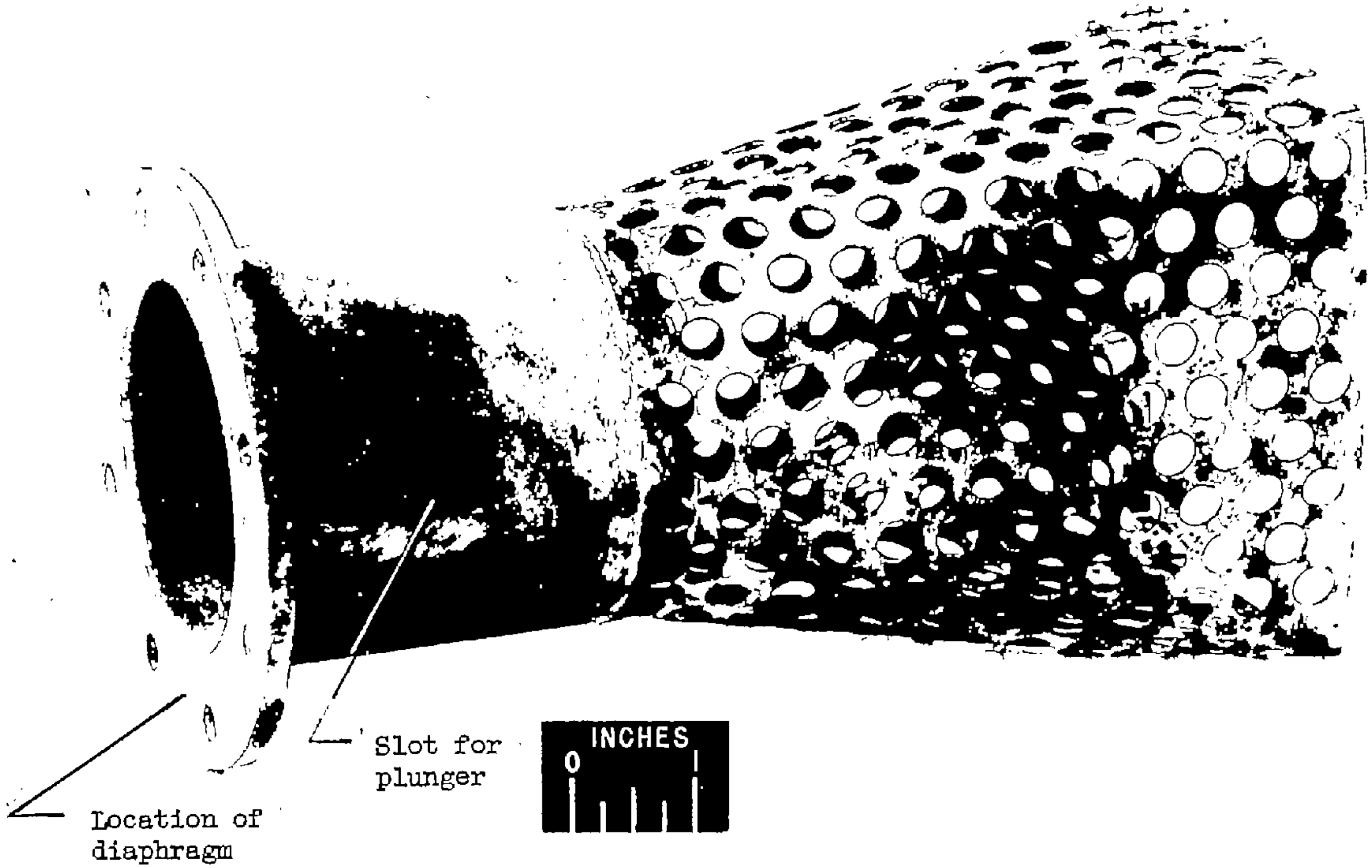

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Figure 3. - Shroud.

Figure 4. - High-speed schlieren photographs of transient flow over wedge using two-dimensional conventional supersonic nozzle.

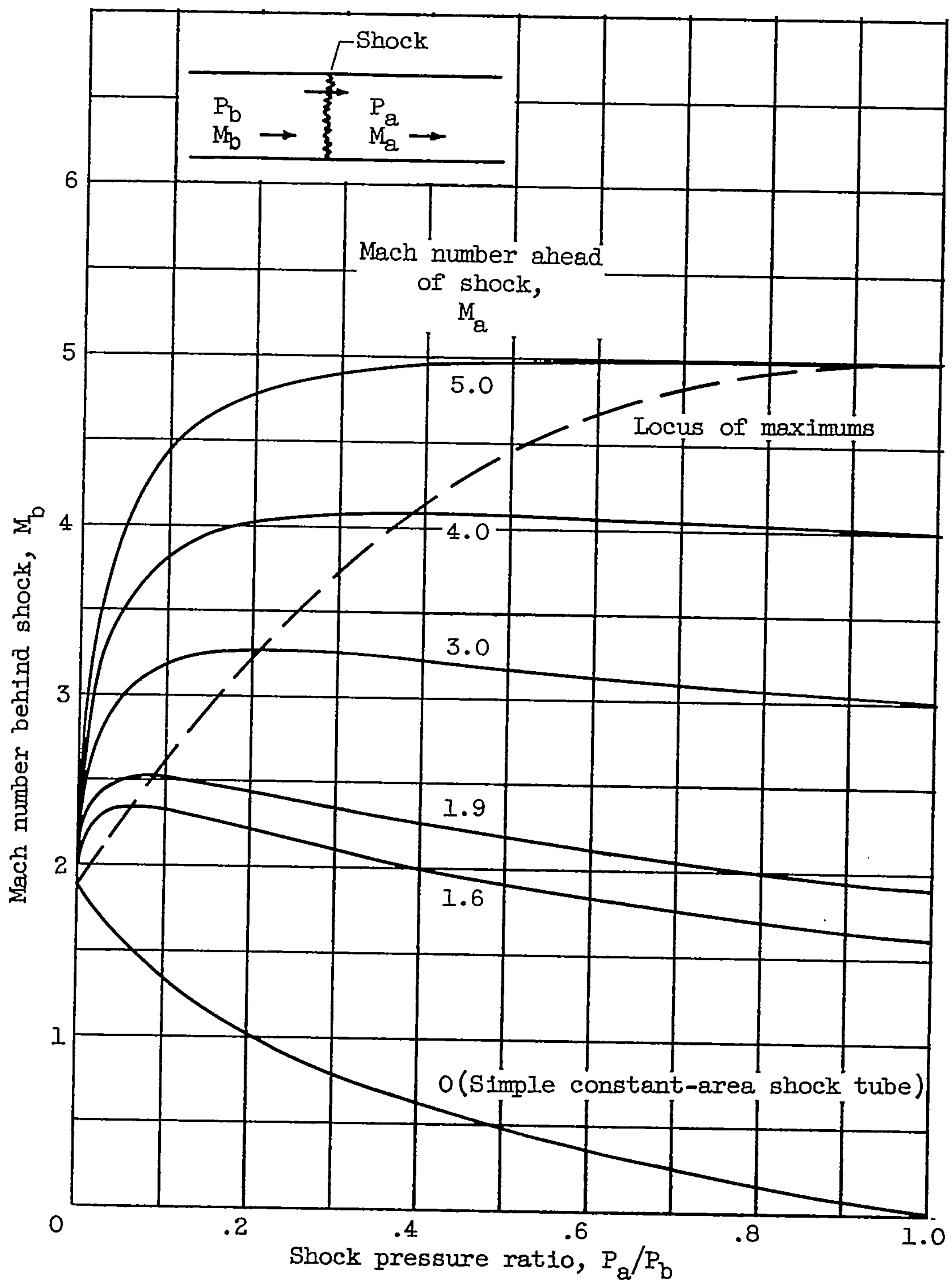

Figure 5. - Theoretical Mach number changes obtainable by passing a shock through a moving stream. Ratio of specific heats, 1.4.

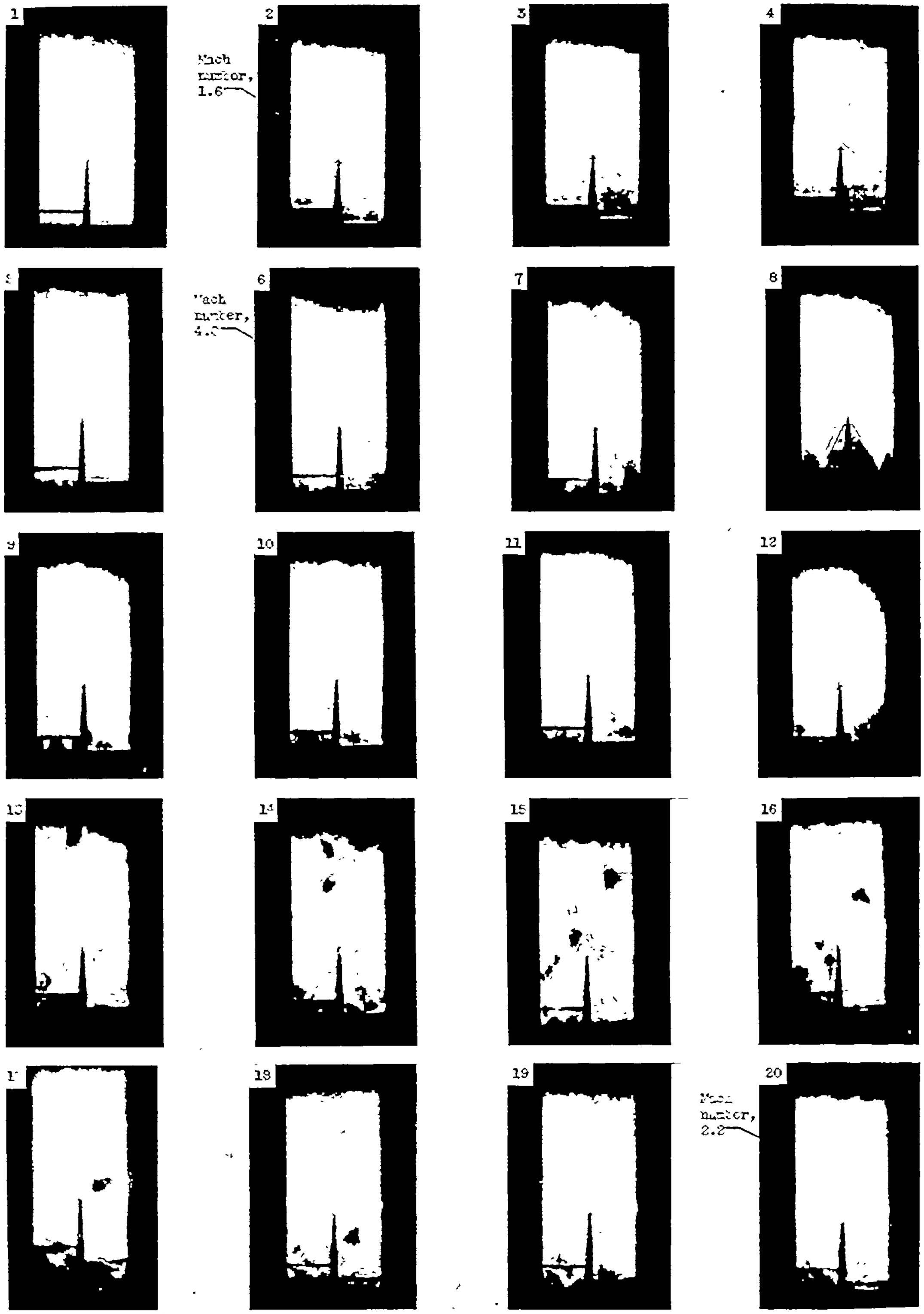

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Figure 6. - High-speed schlieren photographs of transient flow over wedge using axially symmetric multinozzle.